EMI from Spacecraft Docking Systems

Spacecraft Charging ~ Plasma Contact Potentials

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***Abstract*—The plasma contact potential of a visiting vehicle (VV), such as the Orion Service Module (SM), is determined while docking at the Orion Crew Exploration Vehicle (CEV). Due to spacecraft charging effects on-orbit, the potential difference between the CEV and the VV can be large at docking, and an electrostatic discharge (ESD) could occur at capture, which could degrade, disrupt, damage, or destroy sensitive electronic equipment on the CEV and/or VV. Analytical and numerical models of the CEV are simulated to predict the worst-case potential difference between the CEV and the VV when the CEV is unbiased (solar panels unlit: eclipsed in the dark and inactive) or biased (solar panels sunlit: in the light and active).**

Keywords; Orion, CEV, SM, ISS, ESD, spacecraft charging, plasma contact potentials, NASCAP-2K

I. INTRODUCTION

In this paper, the plasma contact voltage (potential difference) between the proposed prototype Orion Crew Exploration Vehicle (CEV) and a docking spacecraft, e.g., a Visiting Vehicle (VV) such as the Orion Service Module (SM) in Low Earth Orbit (LEO) or a malfunctioning satellite in Geosyschronous Orbit (GEO), is first predicted analytically (worst-case) using a simple canonical model of the CEV and the VV. Numerical solutions are also being developed using the NASA/AF Spacecraft Charging Code to refine the predicted results using a high-fidelity model of the CEV and the docking spacecraft at contact. A picture of the proposed prototype Orion CEV on its ejection sled just before loading onto a C17 at the Yuma Proving Ground for a parachute drop test is shown in Figure 1.

Figure 1. Orion CEV Capsule at YPG on its C17 Ejection Sled for Parachute Drop Testing

When the spacecraft docks at the CEV, the potentials of the CEV and the VV could be significantly different due to different spacecraft charging effects on the CEV and the VV prior to docking. Therefore, when the spacecraft docks at the CEV, an intense ESD event could occur, which could disrupt, disable, damage, or destroy sensitive electronic equipment on the CEV and/or the VV.

The CEV can also dock with the ISS in LEO orbit, in which case the Orion CEV would be the VV to the Space Station.

II. INERNATIONAL LOW IMPACT DOCKING SYSTEM (iLIDS)

A. *Androgynous Docking Ring*

As shown in Figure 2, an International Low Impact Docking System (iLIDS), with identical attachments to the CEV and the VV, consists of a circular disk with three embedded permanent magnets, androgynous guiding brackets, and latching mechanisms. The symmetrically placed permanent magnets in the disk produce a concentric magnetic field to slowly attract and align the docking spacecraft on the centerline of the CEV docking ring and to engage the locking mechanisms.

Figure 2. iLIDS Androgynous Docking Ring

B. *Potential Equilibration*

When the two docking rings touch, any potential difference between the CEV and VV equilibrate. It has been thought that, before the two objects come into close proximity and dock, the two metallic objects in the conducting plasma media of the ionosphere would, over the course of time in

orbit, come to similar potentials and there would be no significant potential differences between the CEV and the docking spacecraft. The conducting plasma media in the ionosphere would essentially connect the CEV and docking spacecraft electrically at-a-distance before mechanical contact occurs and would allow the CEV and docking spacecraft potentials to come to a common potential before docking, neutralizing any potential differences, thereby, preventing any severe ESD event from occurring. However, as is shown in this paper, the separate Debye sheaths that develop around the CEV and the docking spacecraft in the plasma ionosphere isolate the two conducting objects in the plasma media, and this equilibration does not occur until the moment of docking.

The contact potential between the International Space Station (ISS) and the Space Shuttle (SS) has been monitored with Langmuir Probes mounted on the ISS on every docking mission. The contact potential has been no greater than ≈ -80 V (approximately half of the solar array bias) on any docking operation and no serious Electromagnetic Interference (EMI) effects have been noted on any of the ISS and/or SS equipment after docking, as shown in Fig. 3.

Figure 3. ISS/SS Docking (Top of Picture)

The new Orion capsule and the Orion service module have not flow in space, and, therefore, have not been tested in orbit. Therefore, the worse-case CEV/VV plasma contact potential at the time of first contact (attachment) is being predicted for the first docking mission.

III. Plasma Contact Potentials

The worst-case plasma contact voltage between the CEV and the docking spacecraft at the time of first contact is being predicted analytically and numerically.

First, a simple analytical model of the docking procedure is developed. Two extreme worst-case scenarios are considered: the CEV and the docking spacecraft are modeled and simulated under sunlit conditions (solar array bias on the CEV) and under unlit shadow (eclipsed) conditions, viz., no photoelectric excitation of the solar panels.

Simple spherical, cylindrical, and rectangular canonical models are developed. For the spherical case, the Debye potential around the "two-body" spherical objects is determined; for the planar case the Debye potential in the plasma sheath between the "two-body" planar objects is determined.

A high-fidelity numerical model of the CEV and VV docking operation is also being developed to predict the plasma potentials using the joint NASA, USAF/AFRL plasma spacecraft charging code: NASCAP-2K. The numerical results will be compared to the analytical results.

IV. Spacecraft Charging

Spacecraft charging, which includes external spacecraft surface charging (especially differential charging) and internal (bulk, buried) dielectric charging, has caused intermittent anomalous behavior of electronic equipment and catastrophic spacecraft failures. Spacecraft charging is due to low-energy plasma currents (charged particle flux to and from the spacecraft from the ambient plasma through which the spacecraft flies) and photoelectric currents under sunlit conditions.

The resultant spacecraft surface potential is the result of the net current flow to/from the spacecraft surface, viz., from plasma electrons and ions impinging on the surface, ejected solar photon-induced photoelectrons leaving the surface, secondary electrons generated by energetic primaries, backscattered electrons (repelled from negative surfaces in shadow) and ions (repelled from positive surfaces in sunlight), and charged particles emitted from the vehicle (active ion emissions ~ plasma contactors). When in eclipse, the spacecraft usually charges negatively due to the attachment of the light, highly mobile, negative electrons relative to the heavy, slowly moving, positive ions. When sunlit, the spacecraft also usually charges negatively due to the photoelectric effect of the solar cells (with a negative bond to the CEV structure), leaving the spacecraft with an excessive of negative charges.

Absolute charging of spacecraft surfaces relative to the ambient plasma is generally not too detrimental; however, differential charging of the spacecraft surfaces, e.g., by surface shadowing, wake effects, etc., and subsequent arc discharges between spacecraft structures, can disrupt spacecraft operations and can cause material damage, create EMI, and produce broad-band transient pulses.

V. Floating Potential

When flying through the plasma, the spacecraft will assume a floating potential different than the ambient plasma potential. The current balance equation in equilibrium for the charging current (conservation of charge) in sunlight or in shadow is

$$I_{total} = +I_{electrons}$$
$$-(I_{ions} + I_{secondary} + I_{photoelectrons} + I_{backscatter} + I_{active}) \cong 0$$

The spacecraft will accumulate charge on its surfaces until an equilibrium condition is reached such that the net current to/from the spacecraft is zero.

As a worst-case condition, the spacecraft is considered to be in eclipse, in LEO orbit (low energy, high-density plasma), with no active sources on-board, and no secondary emissions or back-scattering to reduce the surface electron count.

For a negatively charged CEV structure, the electron and ion currents for simple spherical, cylindrical, and rectangular structures are [1-4]

$$I_{\left\{ e \atop i \right\}} = J_{\left\{ e \atop i \right\}} A_{\left\{ e \atop i \right\}}$$

where, for the repelled electrons

$$J_e = J_{oe} e^{\Delta\mathcal{E}_e}$$

$$J_{oe} = \rho_{oe} v_e = n_e e \sqrt{\frac{kT_e}{2\pi m_e}}$$

and, for the attracted ions

$$J_i = J_{oi} \begin{cases} 1 & \text{(rectangular)} \\ [2\sqrt{\frac{\Delta\mathcal{E}_i}{\pi}} + \mathrm{e}^{\Delta\mathcal{E}_i}\ \mathrm{erfc}(\sqrt{\Delta\mathcal{E}_i})] & \text{(cylindrical)} \\ (1+\Delta\mathcal{E}_i) & \text{(spherical)} \end{cases}$$

$$J_{oi} = \rho_{oi} v_i = n_i e \sqrt{\frac{kT_i}{2\pi m_i}}$$

where

$$\Delta\mathcal{E}_{\left\{ e \atop i \right\}} = -\frac{PE_{\left\{ e \atop i \right\}}}{KE_{\left\{ e \atop i \right\}}} = -\frac{q_{\left\{ e \atop i \right\}}\Phi}{kT_{\left\{ e \atop i \right\}}}$$

and

m_e=9.11x10^{-31} [C]
m_p=1.67x10^{-27} [C] (m_i= $A_i^{'}$ m_p)
e=1.602x10^{-19} [C] (q_e=-e and q_i=+Z_ie)
k=1.38x10^{-23} [J/K]

Note that $A_i^{'}$ is the atomic number of the ion and Z_i is the ionization level of the ion. The areas in the above formulas are the ram side projected area for the slow ions and the total surface area for the fast energetic electrons. The cylinder has transverse and axial projections, and the plane is assumed to be perpendicular to the direction of motion.

At an altitude of ~300 km, the lower limit and worst-case charging environment for arcing on the negatively charged CEV in LEO, the species number densities and temperatures [as determined from the International Reference Ionosphere (ISI2007) data from the Virtual Ionosphere, Thermosphere, Mesosphere Observatory (VITMO)] for atomic Oxygen neutrals (A_i=16) and singly ionized Oxygen atoms (Z_i=1) are

High Density:	Low-Energy (Temp):
n_e=n_i 0.11x10^{12} [1/m^3] n_n 2.6x10^{14} [1/m^3]	$T_e \approx$ 1219[K] = 0.105 [eV] $T_i \approx$ 926 [K] = 0.080 [eV] $T_n \approx$ 734 [K] = 0.063 [eV]

VI. POTENTIAL PREDICTIONS

A. Analytical Solution

From a 1st order worst-case numerical solution (curve fitting, root searching) of the current balance equation (considering only the electrons and ions in eclipse for a negative structure), the spacecraft potential Φ was determined to be on the order of (or slightly less than) the (negative) electron temperature -T_e of the plasma (when expressed in electron volts), i.e., V ≈ -0.5 to -0.7 [V].

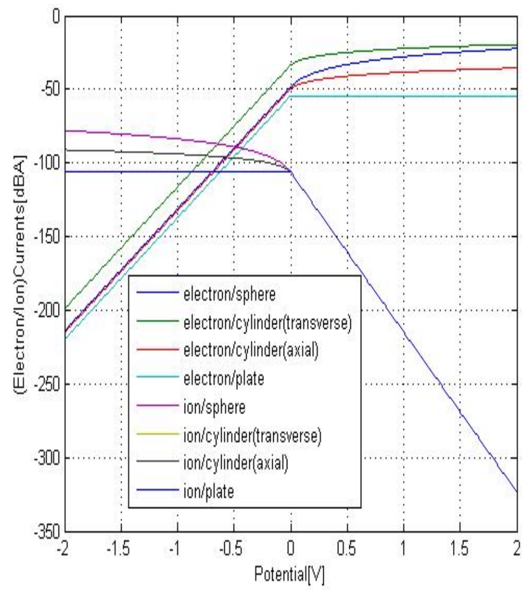

Figure 4. Spacecraft Charging Model (CEV/VV)

B. Numerical Solutions

Biased (sunlit) and unbiased (unlit/eclipsed) conditions of the CEV are considered below. In both situations, worse-case docking conditions are considered, which produce the greatest potential difference between the CEV and the docking spacecraft.

- Plasma Equations

The Vlasov Equation (the collisionless Boltzmann equation) for species s is

$$\frac{D}{Dt} f_s(\bar{r},\bar{v},t) = (\frac{d}{dt} + \bar{a}_s \bullet \overline{\nabla}_v) f_s(\bar{r},\bar{v},t) \cong 0$$

where f_s is the scalar phase-space particle-distribution function of species s.

The "hydrodynamic (spacetime) derivative" is defined as

$$\frac{d}{dt} \equiv \frac{\partial}{\partial t} + \bar{v}_s \bullet \overline{\nabla}_r \xrightarrow{\frac{\partial}{\partial t}=0\ (steady-state)} \bar{v}_s \bullet \overline{\nabla}_r$$

and the acceleration term in the total derivative is the average external Lorentz force per unit mass of species s

$$\overline{a}_s = \frac{\overline{F}_s}{m_s} = \frac{q_s(\overline{E} + \overline{u}_s \wedge \overline{B})}{m_s} \xrightarrow{B=0\ (non-magnetic)} \frac{q_s \overline{E}}{m_s}$$

where

$$\overline{E} \cong -\overline{\nabla}\phi(\overline{r},t)$$

Therefore, the Vlasov equation for species s for steady-state, unbiased conditions reduces to

$$[\overline{v}_s \bullet \overline{\nabla}_r - \frac{q_s \overline{\nabla}\phi(\overline{r},t)}{m_s} \bullet \overline{\nabla}_v] f_s(\overline{r},\overline{v},t) \cong 0$$

For steady-state conditions, the electric field potential satisfies Poisson's Equation

$$\nabla_r^2 \phi(\overline{r},t) = -\frac{\rho_s(\overline{r},t)}{\varepsilon_o}$$

The local solution to the distribution equation for species s is a Maxwellian distribution (with zero drift velocity)

$$f_{so}(\overline{r},\overline{v},t) = A_s e^{-\frac{KE_s}{TE_s}} = A_s e^{-\frac{\frac{1}{2}m_s v_s^2}{kT_s}} = A_s e^{-(\frac{v_s}{v_{so}})^2}$$

where

$$v_{so} \equiv \sqrt{2\frac{kT_s}{m_s}} = \sqrt{2}v_s^{th} = \sqrt{2}\omega_s^p \lambda_s^D$$

One formal solution to the distribution differential equations for species s is a Quasi-Maxwellian distribution (with non-zero drift velocity)

$$f_s(\overline{r},\overline{v},t) = A_s e^{-\frac{KE_s + PE_s}{TE_s}} = A_s e^{-\frac{\frac{1}{2}m_s v_s^2 + q_s\phi}{kT_s}} = f_{so}(\overline{r},\overline{v},t) e^{-\frac{\phi}{\phi_{so}}}$$

where $\phi_{so} \equiv \frac{kT_s}{q_s}$

- Plasma Solutions

The NASCAP-2K "NASA Charging Analysis Program" is being developed to solve the above set of equations simultaneously for the potential and/or temperatures using the finite element method (FEM). The NASCAP-2K code will be used to model the separate docking spacecraft with and without bias on the CEV.

An exploded view of the various sections of the CAD model of the CEV with the iLIDS attached to the top of the capsule is shown in Figure 5. The detailed CAD model of iLIDS is shown in Figure 6. The assembled model is shown in Figure 7.

The numerical results will be compared to the analytical predictions for docking in the dark.

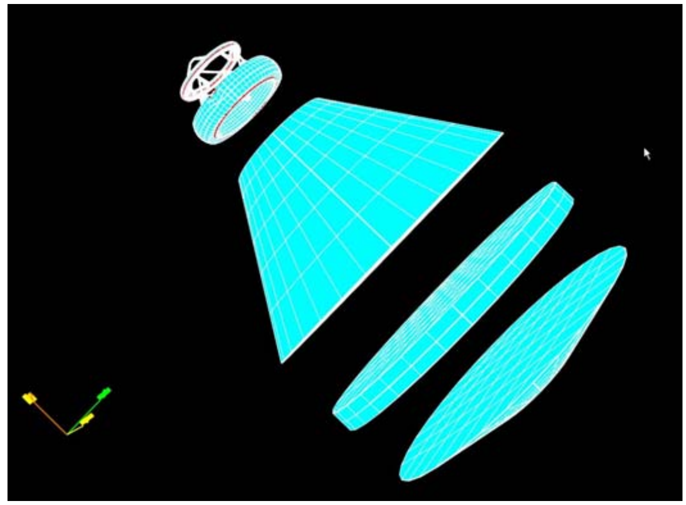

Figure 5. NASCAP-2K CEV Model (Exploded View)

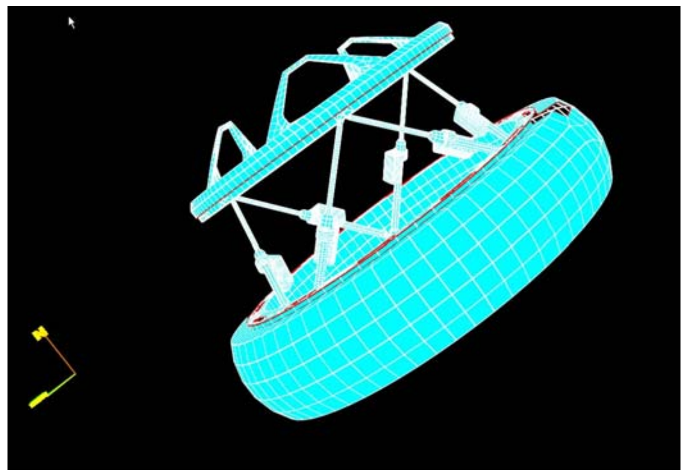

Figure 6. NASCAP-2K iLIDS Model (Detailed View)

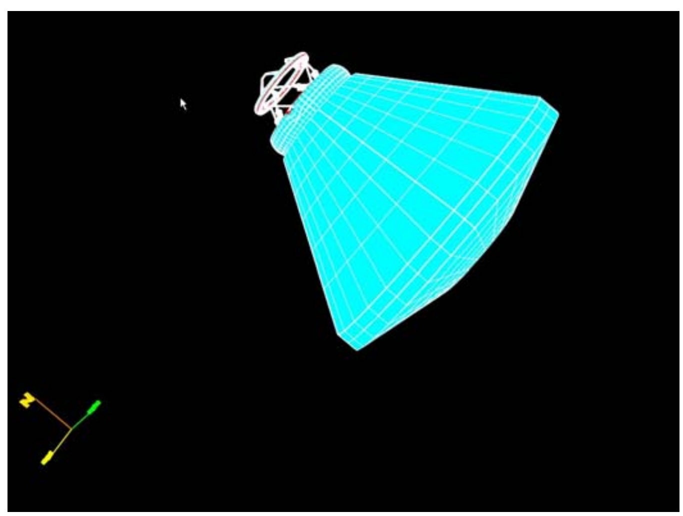

Figure 7. NASCAP-2K iLIDS Model (Assembly)

VII. Preliminary Results

When the CEV us unbiased, with unlit/eclipsed solar arrays, the simple analytical solution for the contact potential predicts results on the order of the electron temperature. The CEV floats at approximately the same potential as the docking spacecraft. Therefore, the voltage between the CEV and the VV is small and it is safe to dock in the dark.

However, when the CEV is biased to a high negative potential, with sunlit solar arrays, there is a larger potential difference between the CEV and the VV. Therefore, when docking in the light, there is the possibility of creating an ESD arc (or a coronal discharge).

No neutralization can occur until the CEV and the VV are within the Debye distance of each other, which is only approximately 7 mm for the worst- case CEV LEO conditions.

VIII. Future Work

NASA is developing the Orion spacecraft capable of reaching high altitudes, such as geo-synchronous orbits. In GEO orbit, the new spacecraft can be used, as an example, to perform satellite repairs. The NASCAP code will also be used to predict the contact potential between the spacecraft and a satellite in GEO.

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